TITANIUM AND ADVANCED COMPOSITE STRUCTURES FOR A SUPERSONIC CRUISE ARROW WING CONFIGURATION

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SUMMARY

Two structural design studies were made, based on current technology and on an estimate of technology to be available in the mid 1980's, to assess the relative merits of structural concepts and materials for an advanced arrow wing configuration cruising at Mach 2.7. Preliminary studies were made to insure compliance of the configuration with general design criteria, integrate the propulsion system with the airframe, and define an efficient structural arrangement. Material and concept selection, detailed structural analysis, structural design and airplane mass analysis were completed for the first study based on current technology. In the second study, based on estimated future technology, structural sizing for strength and a preliminary assessment of the flutter of a strength designed composite structure were completed. In both studies, an advanced computerized structural design system was used, in conjunction with a relatively complex finite element model, for detailed analysis and sizing of structural members.

INTRODUCTION

This paper presents a general description of a study and a summary of results obtained to date by the Boeing Commercial Airplane Company under a contract with the NASA Langley Research Center as a part of the NASA Supersonic Cruise Aircraft Research Program. Detailed structural studies were conducted to establish a realistic metallic aircraft design and an accurate mass estimate for a specific aerodynamic configuration. Further studies are in progress to evaluate potential mass reductions that may be achieved by application of advanced structural concepts and advanced composite materials to the same configuration. A Mach 2.7 arrow wing supersonic cruise configuration was selected for these baseline studies because previous investigations have shown this configuration to be one of the most promising aerodynamic configurations for supersonic cruise applications.

Since supersonic cruise aircraft tend to be large and flexible, aeroelasticity is a major design consideration, and realistic aeroelastic considerations based on analysis of finite-element structural models and sophisticated aerodynamic loading analysis, both steady and unsteady, are required even in a preliminary design study of such a vehicle. The strong interaction of the various disciplines in aeroelastic problems required the use of computer-aided design methods to improve and expedite the aeroelastic and structural resizing cycle (see for example, refs. 1 through 8).

In addition to the usefulness of the results obtained for a specific configuration, these studies provided a unique opportunity to appraise the computer aided design methods, and to identify problems and technology areas requiring further study and development.

CONFIGURATION AND FLIGHT ENVELOPE

During the initial phase of the study, an arrow wing configuration supplied by NASA was analyzed in considerable detail, using criteria and data from the National SST Program, from NASA wind tunnel tests and from an earlier Boeing study of an arrow wing configuration performed under Department of Transportation Contract No. FA-SS-67-3. Refinements were introduced to meet criteria for controllability, stability and performance. To meet minimum safe operational criteria throughout the flight envelope longitudinal and lateral-directional flight critical augmentation systems were incorporated in the design.

An advanced technology afterburning turbojet engine, developed in a study conducted for Langley by the Boeing Commercial Airplane Company (Contract NAS1-11938), was selected for integration with the aircraft. Although this engine definition was considered satisfactory for the structural study, its performance parameters are not representative of current concepts, such as variable cycle engines, both in terms of specific fuel consumption and sideline noise. Thus, no attempt was made to determine the absolute range of the aircraft.

The final configuration used in the structural design study, designated as 969-512B, and the flight envelope are shown in figures 1 and 2, respectively.

STRUCTURAL SELECTION

On completion of the configuration definition, a study of internal structural arrangement was made, utilizing the arrangement developed in the earlier arrow wing study as baseline. Mass differences were evaluated for structural variations, and a multispar arrangement with a small number of ribs, very similar to the baseline arrangement and to that employed on the National SST, was selected for subsequent detail design studies.

Selection of materials based upon current technology for cruising at Mach 2.7 was restricted to those that were proven for primary airframe applications, and Ti-6Al-4V alloy was selected as the primary structural material.

Structural components making the largest contributions to aircraft mass were selected for intensive study. These included wing cover panels, wing internal structure, and the body shell. Many structural concepts were analyzed for mass, manufacturing complexity, stiffness, fatigue, thermal conductance and material cost and assessed qualitatively for maintainability and fail safety requirements. Design loads and environmental conditions were established from the earlier arrow wing study to provide a consistent basis for comparison of concepts. Three locations on the wing and four on the body were chosen for concept evaluation. Wing panel control points and the concepts considered in the concept evaluation process are presented in figure 3.

As shown in figure 4, sandwich panels with titanium face sheets aluminum brazed to a titanium honeycomb core were selected for the entire upper wing surface. The same concept was also selected for the forward, lightly loaded portion of the wing lower surface and both surfaces of the wing tips, outboard of the wing mounted fins. For the heavily loaded portion of the wing lower surface, an integrally machined and welded concept was selected for the portion outboard of the fuselage which carries large tensile loads, while integrally machined waffle construction was selected for the region under the fuselage where large biaxial loads occur. For wing internal structure (spars and ribs), stiffened flat sheet webs were selected for the heavily loaded aft portion and sinusoidally corrugated webs were chosen for the forward portion. Conventional sheet-stringer construction was selected for the fuselage. Initial screening of concepts was governed primarily by mass comparisons; manufacturing complexity, stiffness, maintainability and fatigue contributed significantly to the final selections. In retrospect, after a more thorough evaluation of insulation requirements, it is believed that brazed titanium sandwich would have been a better choice for the entire lower surface of the wing.

ANALYSIS AND DESIGN PROCESS

The computerized system that was used for structural analysis and design was organized around an interim version of the ATLAS Structural Analysis and Design System, interfaced with external programs for flutter analysis and with the FLEXSTAB System for loads analysis. ATLAS is a modular system of computer codes, integrated within a common executive and data base framework, that is operational on the Control Data Corporation (CDC) 6600/CYBER Computers. It was initially developed by the Boeing Commercial Airplane Company and it is currently being extended under a cooperative effort with the NASA Langley Research Center. FLEXSTAB, employed for loads analysis, is a system of programs originally developed under contract to NASA Ames Research Center for stability analysis of elastic airplanes. Modifications to FLEXSTAB to provide loads analysis capability were developed by the contractor in preparation for the arrow wing study.

The major subtasks that make up the structural analysis and design process are identified in figure 5. This may be visualized as three interconnected discipline-oriented segments with the interconnection being provided by the ATLAS system. On the left of the figure is FLEXSTAB used for prediction of steady aeroelastic loads which provides input to the strength design segment shown in the center of the figure. On the right is a group of operations associated with the flutter analysis and design to satisfy flutter criteria. The computer programs performing the various functions are shown in the upper portions of the boxes.

MATHEMATICAL REPRESENTATION OF THE AIRPLANE

Basic data describing the aircraft were developed during the initial stages of the contract, and this information, comprising aircraft geometry, structural arrangement, structural concept, and structural materials, was then used to develop structural, aerodynamic, and mass models of the aircraft to initiate the analysis and design cycle.

Two similar mass models were generated, one for use with FLEXSTAB for loads analysis and the other for use in the flutter analysis. The models differed primarily in the retained node set used. Each model was defined for several total aircraft mass levels. Theoretical masses of structural elements were calculated directly from sizing data. Design studies of local structural details and experience from the National SST Program were then used in converting the theoretical values to actual structural masses.

The structural model of one-half of the aircraft contains approximately 2000 nodes, 4200 elements, and 8500 active degrees of freedom. The sketch in the upper part of figure 6 depicts the complete model. The wing and adjacent fuselage structure are modeled with two-dimensional elements and the remainder of the aircraft simulated with beam elements. Modeling of wing and fin components are shown in other views in the figure. For dynamic analyses a much smaller number of degrees of freedom are retained, 225 for symmetric conditions and 260 for antisymmetric conditions. The complexity of the structural model results from (1) the use of one basic model for both stress and flutter analyses and (2) the detail requirements for meaningful flutter analysis. For the wing, these requirements include structural modeling of the engine support structure (allowing complete motions of the engines), leading and trailing edge controls, wing secondary structure, landing gear and wheel well cutouts, and wing mounted fins as well as primary wing structure. In modeling the remainder of the aircraft, a detailed body idealization is required for wing attachment, while a less sophisticated model is considered adequate for the remainder of the fuselage and the empennage.

INITIAL SIZING AND PRELIMINARY FLUTTER ANALYSIS

Initial structural sizing to start the iterative design cycle was derived from the earlier design study of the arrow wing configuration, with appropriate adjustments to reflect the increase in maximum taxi mass to 340000 kg. This was followed by a preliminary flutter analysis which showed a large deficiency in flutter speed. To insure that airloads and stresses would not be determined for a structure having large flutter deficiencies, the wing tip structure and engine support beams were stiffened to impose realistic stiffness constraints on the strength design process.

LOADS

Load conditions for structural design were based on Federal Aviation Regulations, Part 25, and the Tentative Airworthiness Standards for Supersonic Transports. Loads analysis experience on the National SST Program and the previous study of the arrow wing configuration were used in selecting design conditions. Structural loads were examined for 154 operating conditions, including maneuvers with normal load factors between +2.5g and -1.0g, gust and ground conditions. From these, 25 of the most critical conditions were selected for design. Elastic properties of the structural model with stiffness increases resulting from

the preliminary flutter analysis were used in the FLEXSTAB analysis for determination of static aeroelastic loads.

In addition to airloads, inertia loads and ground loads, several additional factors influencing structural sizing were considered. These included pressurization, fuel containment, acoustic loads, hailstone impingement, lightning strike and thermal effects.

STRENGTH DESIGN OF TITANIUM STRUCTURE

An iterative technique was used in sizing the structure to satisfy strength criteria for the critical load conditions. This required for each resize cycle (1) evaluation of stresses due to the various load conditions, (2) calculation of margins of safety, and (3) determination of changes in member sizing to obtain the desired value of margins of safety. A stability interaction equation was used for evaluation of margins of safety in biaxial compression and shear in the honeycomb sandwich wing covers, and for strength-critical loading conditions a modified Hill's yield criterion was used. A fully stressed design algorithm was used to obtain new member sizes.

Elements in the fuselage were resized by hand, and the resizing process for these elements converged in two design cycles. Lumped areas used in model beam elements in the fuselage are composed in part of effective skin areas, and these are influenced by buckling, body pressurization and thermal stresses. Automated resizing of the fuselage elements was not attempted because of the problem posed by buckled skins and the smaller structural weight savings expected in the fuselage.

Elements in the wing covers were resized using an automated resizing module with convergence, as measured by total mass change, occurring in three cycles. Successive change in total face sheet thickness and theoretical wing weight are displayed in figures 7 and 8, respectively. In figure 7, three sets of gages are shown for selected panels. Reading from top to bottom, the first set of values are the initial values of upper/lower surface panel gages. The second and third sets of gages were obtained from successive cycles of automated resizing. For lower surface panels of integrally stiffened skin construction, the initial value is the area per cm of skin plus stiffener while the second value, in parentheses, is the skin gage. For upper and lower surface panels of honeycomb sandwich construction, the single value shown is the sum of inner and outer face sheet gages. Margins of safety were calculated considering stability, material strength (or allowable stress level), and fail safety for multiple load cases. The loading components included membrane stress resultants due to overall load condition, bending due to

local pressure loading, and thermally induced loads. Minimum gage constraints were based on foreign-object damage and acoustic effects. Constraints relating the sizing of adjacent elements, such as maintaining cap areas of at least one-quarter of the area of the largest adjacent panel for fail safety, were manually imposed between resize cycles. Conditions governing wing cover thicknesses in the strength design are shown in figure 9.

FLUTTER ANALYSIS AND SIZING OF TITANIUM STRUCTURE FOR STIFFNESS

Engineering judgment, based on experience on the National SST Program, was used in defining structural design changes to meet flutter criteria. The starting point for stiffness redesign was the strength-designed structure; no reductions in gages or member sizes, below the values specified for strength, were allowed.

A condition of symmetric flutter at M = .9 and heavy gross mass was selected for analysis to evaluate effects of design changes on the flutter margin. This was confirmed later, as a satisfactory approximation to the critical design condition, by conducting symmetric and antisymmetric flutter analyses of the final design at high and low masses for a range of subsonic and supersonic Mach numbers. Flutter speeds calculated for the critical condition at several stages in the structural design process are shown in figure 10.

The stiffness modifications of the initial structure, prior to strength sizing, consisted of increases in stiffnesses of nacelle support beams to 4.5 times the initial value, an increase in low-speed (outboard) aileron cover thickness by a factor of 4.0, a two fold increase in spar and wing-cover thicknesses outboard of the wing-mounted fin, and addition of high-speed control locks to the low-speed aileron and outboard flaperon. After strength resizing, with a slight decrease in flutter speed, the effect of wing-tip stiffening and control locks were evaluated individually, and they were then retained in subsequent analyses.

A total of nine flutter analyses were made in establishing the final stiffness design. In addition to increasing the stiffness of individual structural members the maximum wing thickness ratio was increased from 2.8% to 3.5% at the fin station with the increment decreasing linearly to zero at the wing tip and at the outboard nacelle station. Then a complete analysis of the final configuration was made for symmetric and antisymmetric flutter at high and low gross masses for subsonic and supersonic conditions.

The mass increase associated with structural changes and the mass equivalent of the drag increase due to the wing-tip thickening is 4640 kg. It was concluded that further effort to increase the flutter speed by structural changes based on engineering judgment would produce an unrealistically high weight penalty. Hence, the subsonic dive placard, V_D, was reduced by 93.6 km/hr, imposing a range decrease of 40 km with fixed fuel loading, or an increase of 600 kg in fuel load required for constant range. Undoubtedly a significant mass reduction and/or a higher placard could be achieved by using formal optimization techniques. However, that approach could not be implemented within the scope of the study.

STRUCTURAL CONCEPTS FOR ADVANCED COMPOSITES

The candidate advanced composite structural concepts (honeycomb sandwich, sheet stiffener and stiffened thin honeycomb sandwich) are shown in figure 11. Comparisons of mass per unit area for each of the concepts, in borsic-aluminum, shows that the conventional honeycomb sandwich panel has the lowest mass. (It should be noted that three of the panels have been evaluated with two different allowable shear stresses. The lower values were based on preliminary published data. Unpublished test data were obtained through consultation with NASA personnel, which provided justification for higher allowable shear stresses.) A further mass comparison between borsic-aluminum sandwich and graphite-polyimide indicated the latter was the lighter of the two.

Thermal insulation of the fuel is a critical design consideration since the fuel is used as a heat sink. The conventional sandwich panel requires the least additional thermal insulation of the three concepts considered. The honeycomb sandwich panel also offers the least fuel vapor ignition hazard from lightning strike. Based on this evaluation, the honeycomb sandwich panel concept was selected.

MATERIAL SELECTION

Projected properties of candidate material systems for the year 1986 are shown in table 1. It was assumed in this study that matrix development will resolve the microcracking problem of transversely loaded lamina so that cross-plied laminates will be fiber critical. During the material selection process, only balanced symmetrical laminates were considered. Based on a comparison of specific strengths and stiffnesses, high strength graphite polyimide and boron polyimide were selected for further study. As a result of comparative mass analyses of panels designed for representative loading conditions high strength

graphite polyimide was finally selected for use in honeycomb sandwich panel face sheets.

Approximately 50% of the wing surface was minimum gage in the titanium design. This percentage will increase for the composites with polymer matrix materials. Estimated minimum gages of tapes to be available by 1986 and a mass per unit area comparison are shown in table 2.

WING SKIN PANELS RESIZED FOR ADVANCED COMPOSITE

To evaluate mass reductions that may be achieved by application of advanced composite materials, cover panels of the primary wing box were replaced with sandwich panels of graphite/polyimide for detailed analysis. The internal titanium rib and spar structure was not revised. Balanced, symmetric laminates were utilized in panel face sheets to avoid post-cure warping and to simplify analysis.

The wing surface panels of the main wing box were divided into 16 zones for generation of preliminary sizing input (see figure 12). Each zone generally contained panels having the same layup and subjected to similar spanwise, chordwise and shear load components or similar constraint conditions, such as minimum gage. For the initial input layups were estimated for the load intensities determined in the earlier analysis of the titanium structure.

The ATLAS design module resized the panels based only on allowable material properties, since buckling is not presently included in automated analysis and design capability. For this reason, the materials designated for the upper surface are distinct from those designated for the lower surface to permit the use of reduced allowable strains to satisfy the buckling requirement. Stability analyses were performed separately with the Boeing-developed COOPB program.

Figure 13 illustrates the normalized theoretical weight changes in the wing box primary structure for successive changes in the wing box primary structure for successive resize cycles. The first two resizes were performed with only the lower bound restriction that the laminae in any one of the orientation angles (0, $\pm\pi/4$ and $\pi/2$) could not disappear. In the limit this would require four laminae per face sheet. In the third resize cycle the minimum gage constraint previously specified for graphite polyimide face sheets was imposed. The weight increment from the second to the third resize indicates that a significant penalty is involved in satisfying the minimum gage constraint.

Figure 14 shows some panel sizing results from the ATLAS automated resizing. The upper values for a given panel represent the initial sizing, with the upper surface sizing listed on the left and lower surface sizing on the right. The sizing notation corresponds to the Standard Laminate Code (e.g. 16/8/8 describes the laminate $[0_{16}/\pm\pi/4_8/\pi/2_8]$). The 0 direction is parallel to the spar at the aft edge of the element. When the inner and outer face sheets for either the upper or lower panel have different sizing, the two values are shown within a brace. These are easily distinguished since the inner face sheet is always equal to or less than the outer face sheet for a given panel. Reading from top to bottom for a given panel, the consecutive values are the initial sizing, the sizing after the first resize, the sizing after the second resize and the final sizing which satisfies strength and minimum gage constraints. If a fourth value is not listed, the third value satisfies strength and minimum gage constraints.

With the exception of the lower surface just inboard of the outboard engine beam, the panels adjacent to the rear spar are predominantly unidirectional laminates oriented parallel to the rear spar.

Along the side-of-body on the lower surface, body bending induces chordwise loads that peak inboard of the wheel well where up to six $\pi/2$ plies are required.

The largest requirement for $\pm\pi/4$ laminae occurs 6 spars forward of the rear spar midway between the engine beams on the upper surface. Note that the lower panel does not require the $\pm\pi/4$ plies.

Flutter Appraisal of Advanced Composites Strength Design

The thickened wing tip, locked low speed controls, stiffened engine beams with diffusion ribs and stiffened wing rear spar which were developed during stiffness redesign of the metallic aircraft design were imposed as constraints during strength resizing of the advanced composite cover panels.

The flutter appraisal technique, based on energy balance at neutral stability and engineering judgment, that was utilized for the metallic aircraft study is also being used for the advanced composite study. The predominantly spanwise laminates, which characterize the strength design in advanced composites, result in a relatively low flutter speed and frequency compared with the equivalent metallic strength design as shown in figure 15. A comparison of modal frequencies and dominant contributions to the energy balance at neutral stability is shown in table 3. The first significant mode shape change appears for mode 6 where increased wing tip torsion with the advanced composite cover

panels is responsible for considerably greater energy extraction from the airstream.

Stiffness Redesign of Composite Cover Skins

Two approaches have been explored to satisfy flutter criteria in the advanced composite study. In the first approach, the balanced, symmetric (orthotropic) composite layup philosophy used in the strength design was preserved but sufficient $\pm\pi/4$ plies were added to increase wing torsional stiffness in the heavily loaded aft wing box, such that the effective shear modulus of a representative composite panel layup is equal to one-half the shear modulus of the corresponding titanium panel. This resulted in a 13/9/1 layup. Finally, both bending and torsional stiffness were increased in the wing tip by adopting a 12/8/8 layup, which provides the equivalent of one-half the titanium panel stiffness design in the earlier study.

The second stiffness redesign strategy is intended to exploit the potential advantages of an anisotropic layup in the wing tip region as discussed by Austin et al. in reference 9.

The general character of this study is illustrated in figure 16, in which the twist coupling coefficient, η, is plotted versus effective shear modulus for various orthotropic and anisotropic cover layups. The twist coupling coefficient is considered positive when nosedown rotation of a streamwise wing section is induced by upward loading (positive washout). Effects of two types of anisotropy are illustrated -- (1) inequality of $+\pi/4$ and $-\pi/4$ lamina; (2) orientation other than 0 of the initially listed lamina, as indicated by subscripts. Favorable effects on flutter speed are generally associated with decreasing η and increasing shear modulus. Compared with the 12/8/8 wing tip layup, an increase of about 22% in effective shear modulus and a decrease of 15% in η is achieved with a $15_{+\pi/2}/10/1$ layup (spanwise fibers oriented $\pi/12$ rad. aft).

Calculated flutter speeds for the stiffened orthotropic and the anisotrophic layup are 173 and 175 m/sec, respectively, at M = .9. The flutter frequency is 1.62 Hz for both cases. In order to obtain the required flutter speed (see figure 15), it probably will be necessary to further stiffen the structure as described above to the level of the metallic aircraft. Also, it may be advantageous to employ fibers with properties that are intermediate between the values for high strength and high modulus fibers.

CONCLUDING REMARKS

An in-depth structural design study of an arrow wing supersonic cruise aircraft has been completed utilizing structural materials and concepts that are representative of the current level of technology. This part of the study has provided a baseline aircraft design for application and evaluation of advanced technology, such as composite structural materials and active controls.

The analysis and design of the composite wing shell has provided a successful demonstration of automated design capability for application of an advanced composite material to a complex structure. It is expected that an all composite structure will permit a greater reduction in structural mass than the combination of metal and composite materials that was used in the current study. Because of the importance of aeroelastic requirements, particularly flutter prevention, further consideration should also be given to the development of optimum fiber properties, intermediate between the high strength and high modulus fibers that were considered here. Unbalanced and unsymmetrical laminates should also be explored further for potential benefits in the solution of aeroelastic problems and for manufacturing feasibility.

The following conclusions are considered generally applicable to the structural design of large supersonic cruise aircraft, irrespective of the choice of material:

(1) An integrated design system should be used in the preliminary design phase.

(2) Static aeroelastic effects and flutter should be considered as early as possible in the design process.

(3) Automated modeling methods and sophisticated graphics capability are desirable to decrease manpower and flow time for generation and validation of the structural model.

(4) Automated resizing for strength, using unrefined initial estimates of member sizes, is an important factor in reducing design cycle time.

These points are discussed at some length in reference 8.

Experience gained from this study has identified the following problem areas in basic technology where further work is needed:

(1) Use of Mechanical Fasteners in Composite Materials

Design and development effort is needed to investigate new and innovative methods for efficient transfer of high concentrated loads.

(2) Standard Test Specimens and Test Procedures for Composites

Work is needed to develop standard test specimens and test procedures and to define the relationship between basic material properties and the strength of structural elements.

REFERENCES

1. Thomas, R. M., et al.: "Aircraft Strength and Stiffness Design Automation," U.S.-Japan Design Automation Symposium, Tokyo, Japan, August 21-22, 1975.

2. Giles, G. L.: "Procedure for Automating Aircraft Wing Structural Design," Transactions of the ASCE, Journal of the Structural Division, January 1971, pp. 99-113.

3. Giles, G. L.; Blackburn, C. L.; and Dixon, S. C.: "Automated Procedures for Sizing Aerospace Vehicle Structures (SAVES)," Journal of Aircraft, Vol. 9, No. 12, Dec. 1972, pp. 812-819.

4. Heldenfels, R. R.: "Automating the Design Process: Progress, Problems, Prospects, Potential," AIAA Paper 73-410, Williamsburg, Virginia, 1973.

5. Miller, R. E., Jr.; and Hansen, S. D.: "Large-Scale Structural Analysis of Current Aircraft," Symposium on Large Computations, ASME, New York, 1970.

6. Swan, W. C.; and Kehrer, W. T.: "Design Evolution of the Boeing 2707-300 Supersonic Transport; Part I, Configuration Development, Aerodynamics, Propulsion, and Structures; Part II, Aircraft Design Integration and Optimization," Presented at the 43rd Meeting of the Flight Mechanics Panel of AGARD, Florence, Italy, October 1-4, 1974. AGARD Conference Proceedings No. 147.

7. Turner, M. J.; and Bartley, J. B.: "Flutter Prevention in Design of the SST," Symposium on Dynamics of Structures, Stanford University, 1971.

8. Robinson, J. C.; Yates, E. C., Jr.; Turner, M. J.; and Grande, D. L.: "Application of an Advanced Computerized Structural Design System to an Arrow-Wing Supersonic Cruise Aircraft," AIAA Paper 75-1038, Los Angeles, California, 1975.

9. Austin, F.; Hadcock, R.; Hutchings, D.; Sharp, D.; Tang, S.; and Walters, C.: "Aeroelastic Tailoring of Advanced Composite Lifting Surfaces in Preliminary Design," Proceedings of AIAA/ASME/SAE 17th Structures, Structural Dynamics and Material Conference, King of Prussia, Pennsylvania, 1976.

TABLE 1.- 1986 ADVANCED-COMPOSITE DESIGN ALLOWABLES, "B" VALUES

		HIGH STRENGTH GRAPHITE/POLY	HIGH MODULUS GRAPHITE/POLY	BORON/ POLYIMIDE	BORSIC/ ALUMINUM
TENSILE STRENGTH	GPa	2.03	1.020	1.344	1.344
COMPRESSIVE STRENGTH	GPa	2.00	.869	2.41	2.43
TENSILE MODULUS	TPa	0.1379	.276	.221	.221
COMPRESSIVE MODULUS	TPa	0.1379	.276	.221	.221
TENSILE STRAIN	μm/m	14,750	3,700	6,100	6,100
COMPRESSIVE STRAIN	μm/m	14,500	3,150	11,000	11,000
DENSITY	kg/m^3	.1550	.1605	.2010	.2710
FIBER VOLUME FRACTION		.60	.60	.50	.50

TABLE 2.- MINIMUM GAGE CONSIDERATIONS

MATERIAL	UPPER SURFACE			LOWER SURFACE		
	SKIN GAGE (mm)		MASS kg/m^2	SKIN GAGE (mm)		MASS kg/m^2
	INNER	OUTER		INNER	OUTER	
TITANIUM	.254	.381	2.81	.254	.508	3.37
H/S GRAPHITE POLYIMIDE $(0/\pm45/90)_S$	.406	.610	1.577	.406	.813	1.889
BORON POLYIMIDE $(0/\pm45/90)_S$	.925	.925	3.71	.925	.925	3.71

BASED ON:

MINIMUM GAGE OF TAPES AVAILABLE BY 1986

GRAPHITE POLYIMIDE .051 mm/PLY

BORON POLYIMIDE .132 mm/PLY

MINIMUM GAGE FOR PRACTICAL CONSIDERATIONS

GRAPHITE POLYMIDE

.076 mm/PLY UPPER SURFACE OUTER SKINS

.102 mm/PLY LOWER SURFACE OUTER SKINS

TABLE 3.- STRENGTH DESIGN COMPARISON FOR FLUTTER

MODE	FREQUENCY (Hz)		ENERGY CONTRIBUTION AT NEUTRAL STABILITY (SOURCE POSITIVE)	
	METALLIC	COMPOSITE	METALLIC	COMPOSITE
PLUNGE	0.	0.	-.082	-.075
PITCH	0.	0.	-.104	-.138
1	0.98	0.80	-.620	-1.0
2	1.19	0.97	-1.0	-.415
3	2.16	1.82	.854	.637
4	2.40	2.00	-.167	-.551
5	2.77	2.69	.039	-.152
6	3.12	2.89	.138	.606
7	3.39	2.99	.613	.438
8	3.80	3.30	-.004	.016
9	4.11	3.55	.002	.142
10	4.85	3.86	-.023	.061
11	4.94	4.63	-.024	-.041
12	5.77	5.04	.267	.267
13	6.23	5.43	-.003	-.006
14	6.62	5.66	.165	.066
15	6.87	5.78	-.013	.170
16	7.64	6.51	-.004	-.008
17	7.84	7.36	-.013	-.009
18	8.39	7.40	-.013	-.010

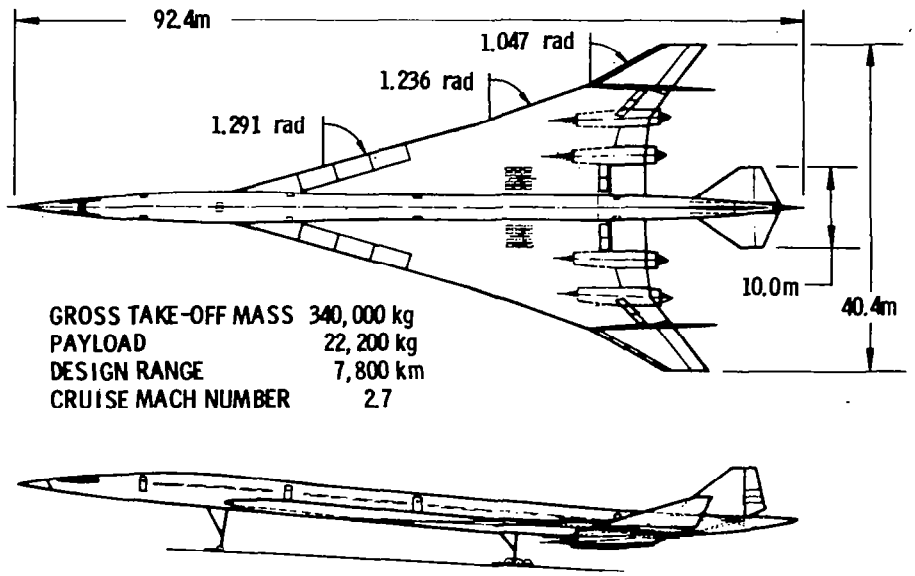


Figure 1.- Configuration - model 969-512B.

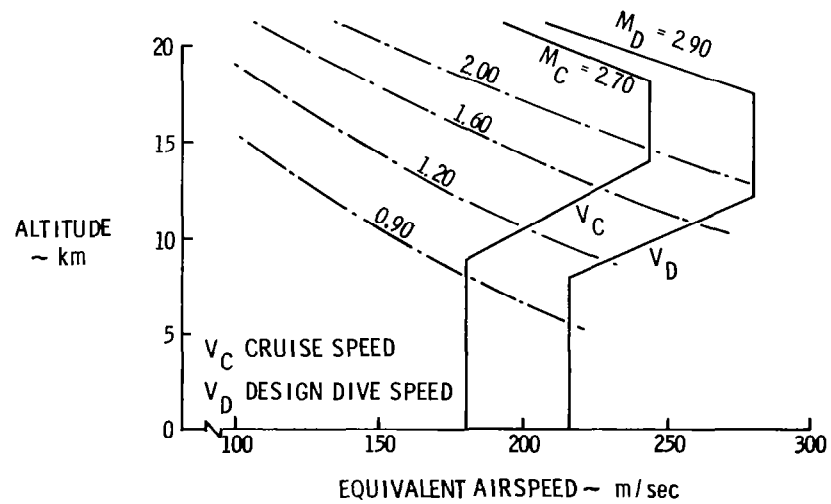


Figure 2.- Flight envelope.

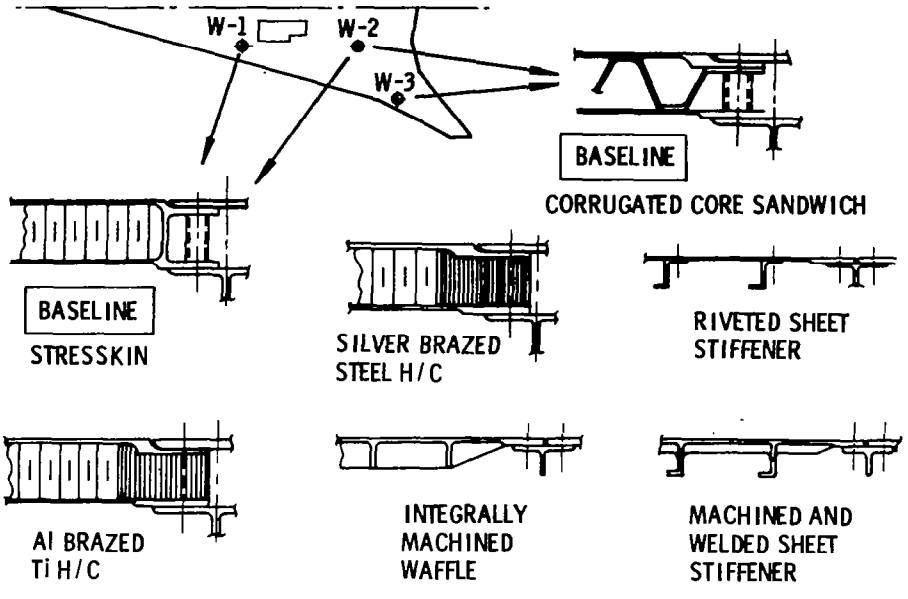


Figure 3.- Wing panel structural concepts.

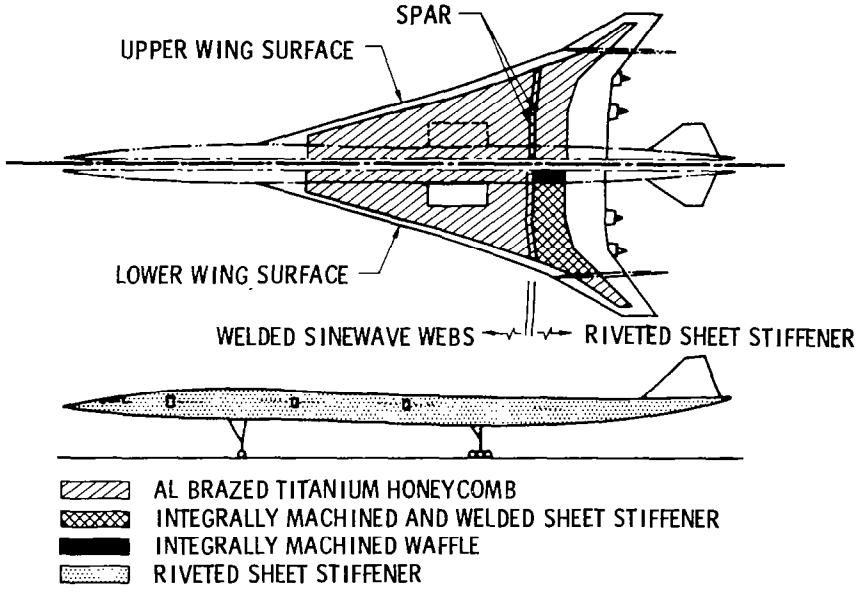


Figure 4.- Wing and body structural selection.

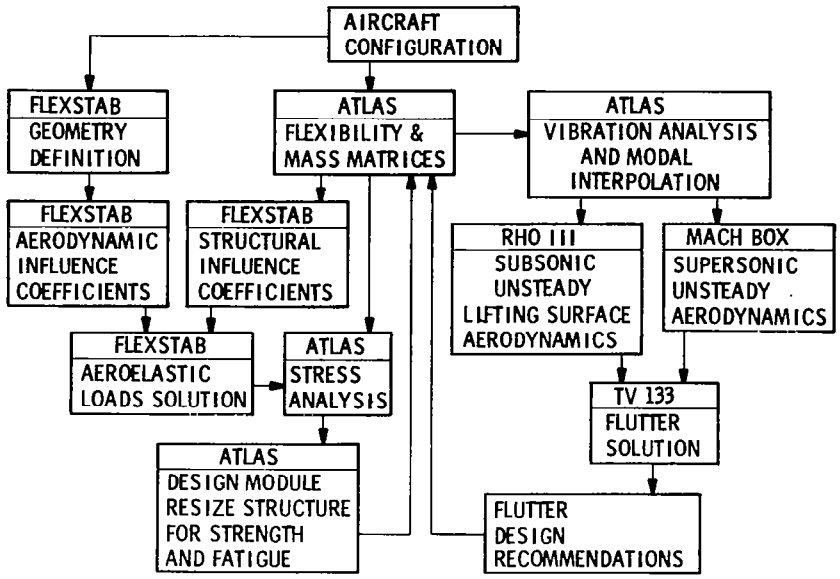


Figure 5.- Analysis and design process.

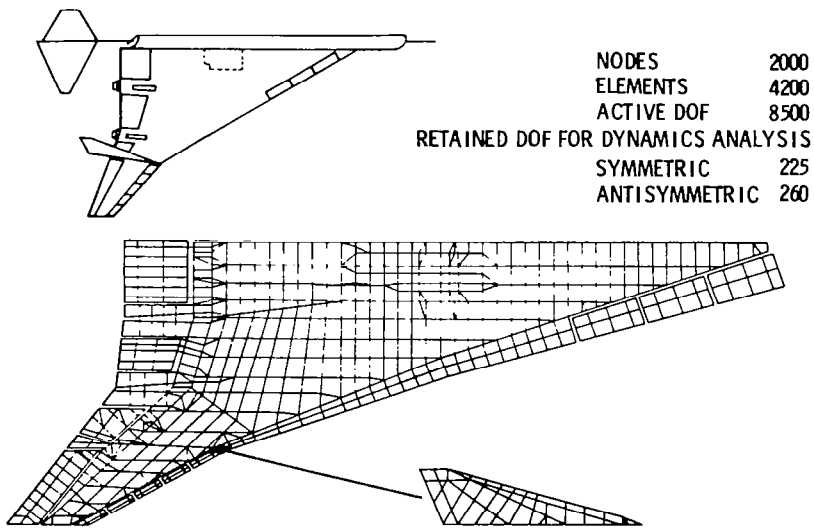


Figure 6.- Finite-element model.

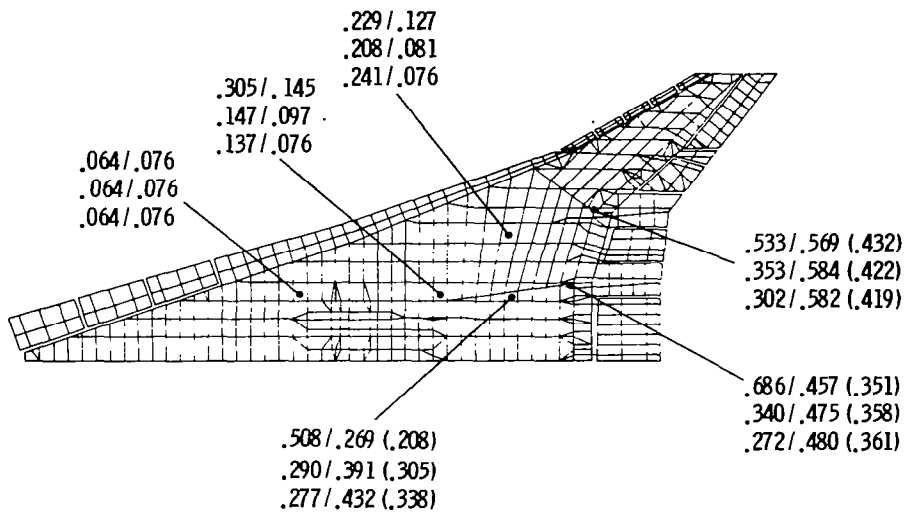


Figure 7.- Structural changes due to strength resizing.

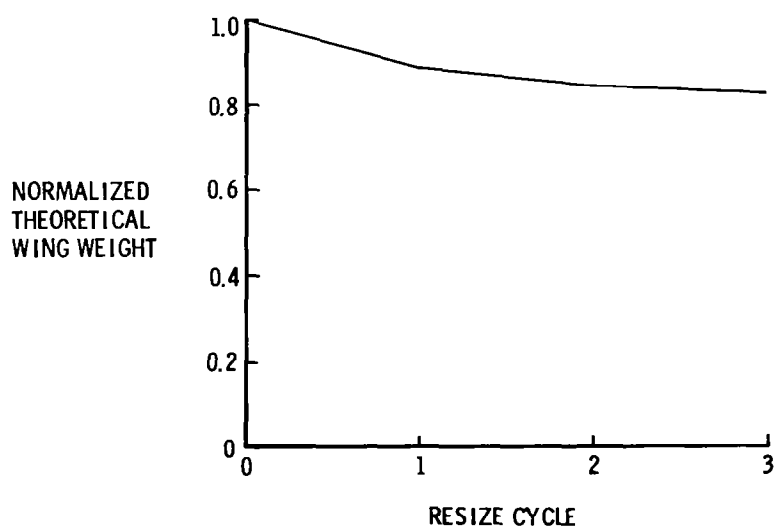


Figure 8.- Theoretical wing weight from ATLAS resizing.

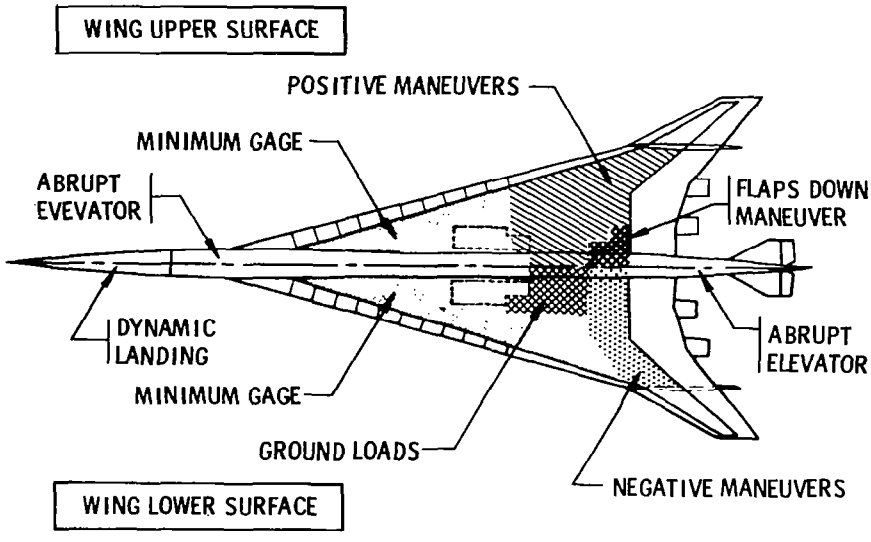


Figure 9.- Critical design conditions.

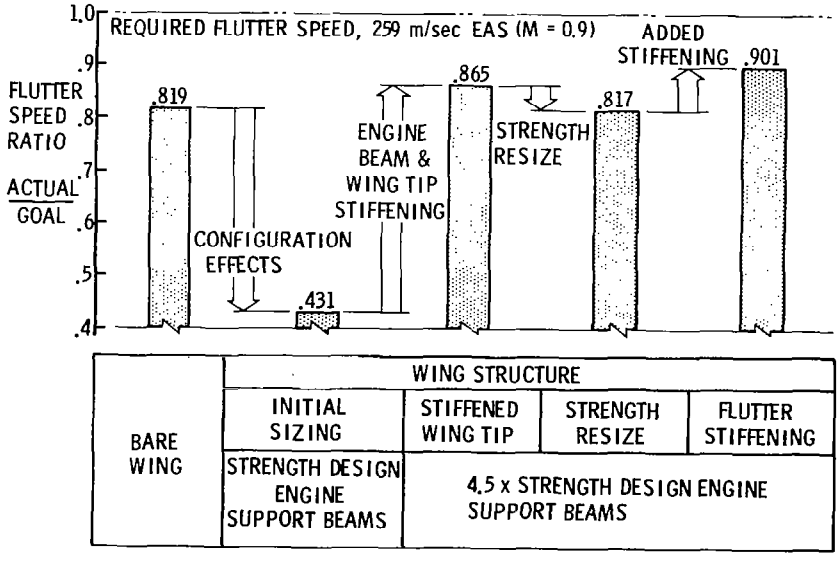
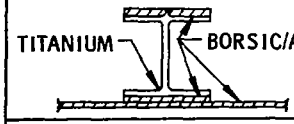
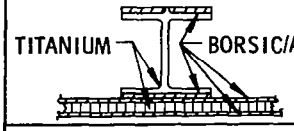
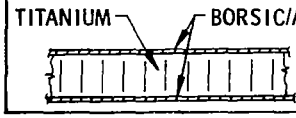


Figure 10.- Effect of structural changes on critical flutter speed.

STRUCTURAL CONCEPT	UNIT MASS, KILOGRAMS PER SQUARE METER		
	BODY	WING UPPER SURFACE	WING LOWER SURFACE
	21.1	** 20.9	20.7 16.89*
	20.9	19.29 16.45*	** 21.5
	17.63	** 17.63	19.58 16.7*

INDICATED MASSES DO NOT INCLUDE THERMAL INSULATION

* HIGH SHEAR ALLOWABLE DERIVED FROM NASA LRC TESTS

** NOT ANALYZED WITH HIGH SHEAR ALLOWABLE -- SIMILAR COMPARISONS EXPECTED

Figure 11.- Structural concept selection - mass comparison.

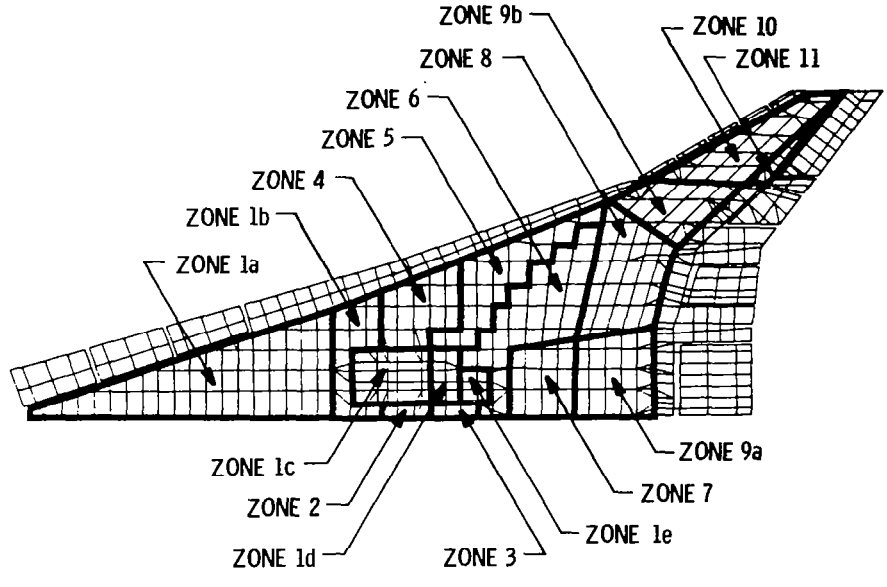


Figure 12.- Zones used for resize.

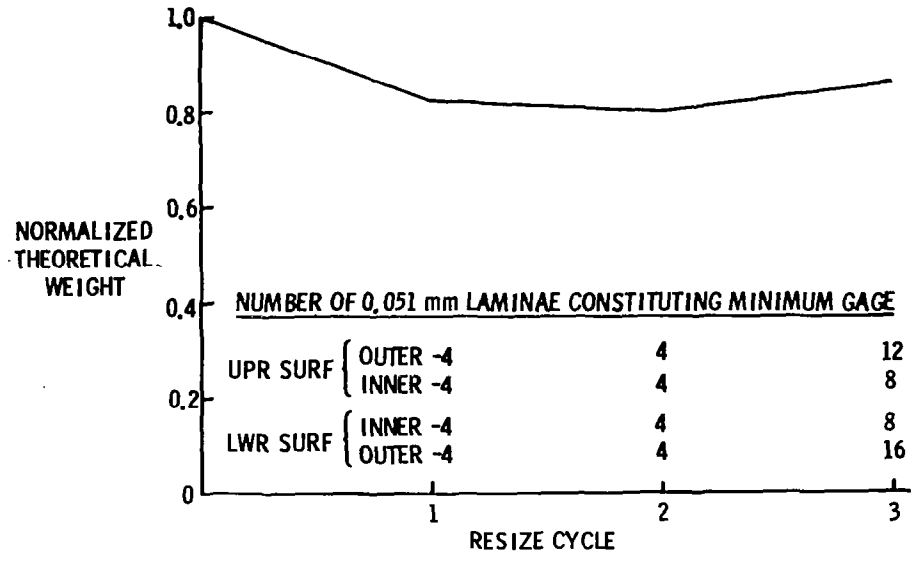


Figure 13.- Theoretical weight changes for wing-box primary structure from ATLAS resizing.

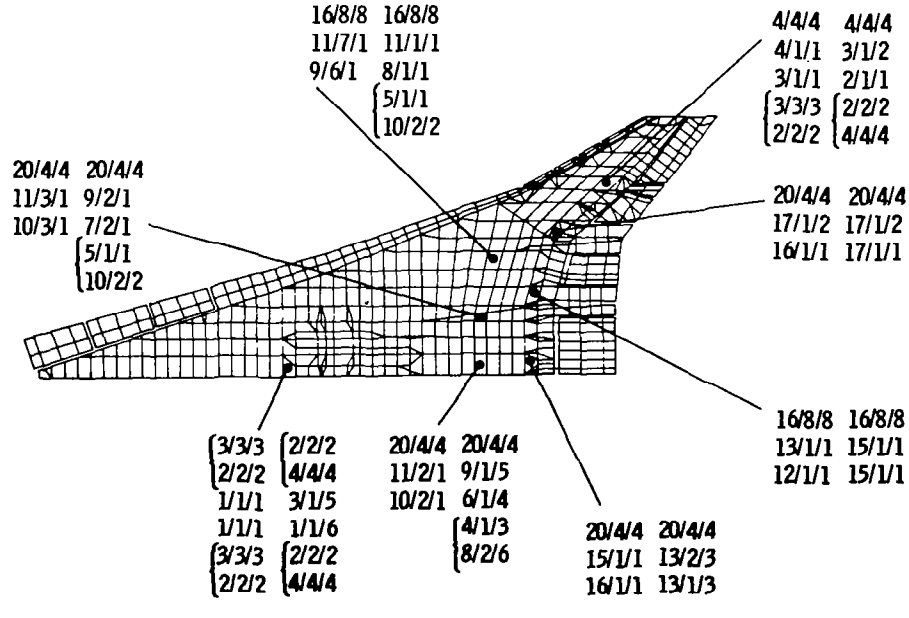


Figure 14.- Typical wing panel resizing.

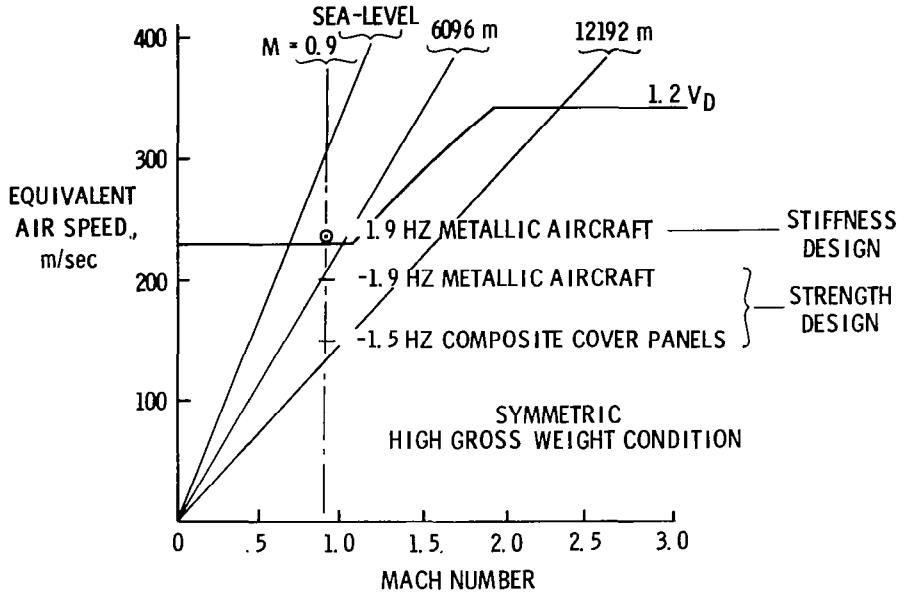


Figure 15.- Comparison of flutter speeds.

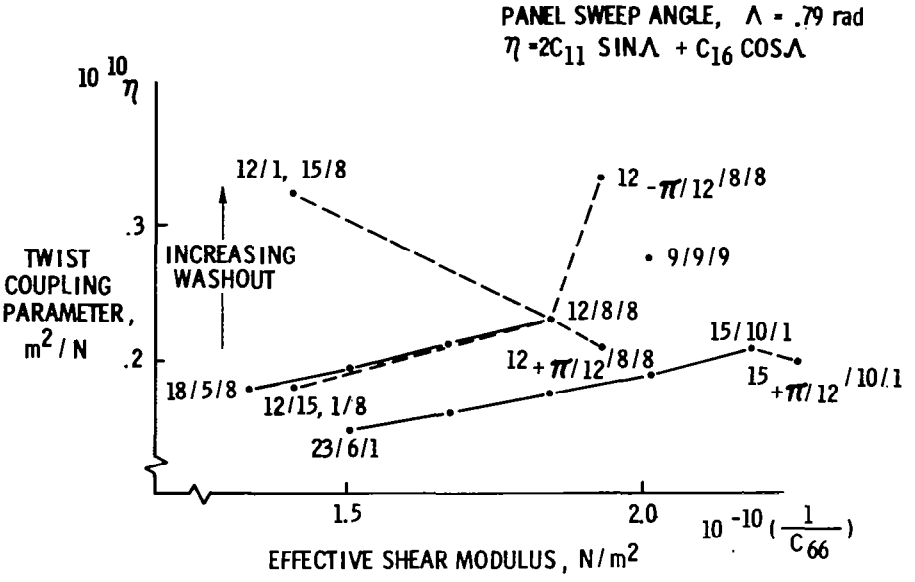


Figure 16.- Anisotropic coupling trends with high-strength graphite polyimide.